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One Dimensional Model of a Solar-Thermal Thruster Using a Porous Absorber/Heat Exchanger*

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October 15, 1993

Abstract

This paper presents a one dimensional model of a solar-thermal thruster using a porous disk as a concentrated-sunlight absorber and heat exchanger. This type of thruster is one of several high *Isp* thruster concepts being studied by the Phillips Laboratory to transfer payloads from low earth orbit to geosynchronous orbit. Solar propulsive techniques should deliver close to 1000 *sec* of *Isp* using a very low molecular weight propellant, hydrogen. The purpose of this model is to determine the dependency of thrust, specific impulse, and energy conversion efficiency upon a variety of parameters including solar concentration, and propellant mass flow rate. This model incorporates a primary concentrator, a window to admit concentrated light and keep propellant from escaping to the vacuum, a secondary nonimaging concentrator, a porous disk as an absorber/heat exchanger, and a nozzle. The model assumes that the light intensity anywhere within the porous absorber/heat exchanger obeys the diffusion equation with radiant energy being exchanged with the porous material's thermal energy. The propellant is an ideal gas which undergoes throttling and collects heat convectively as it passes through the porous absorber/heat exchanger. The specific impulse and thrust are calculated using simple one dimensional nozzle theory and the propellant temperature delivered by the heat exchanger. Future extensions to this model will include hydrogen properties derived from its partition function, correction for nondiffusive radiation transfer in the porous media, and extension of the model to two dimensions. A number of absorber/heat exchanger experiments are planned for the next year. These will flow propellant through porous carbon-carbon disks of various pore densities at a variety of flow rates. The results will guide model improvements.

Introduction

The use of sunlight as a propulsion energy source is attractive because it allows a wider choice of propellant. This is because the propellant is not constrained to be a byproduct of a chemical reaction. Hydrogen may be used, with no oxidizer, so that the *Isp* is maximized [1]. Thermodynamics allows temperatures approaching that of the solar surface (about $5800K$)[2]. Therefore, the propellant temperature is limited only by the material properties which heat and contain the propellant. Refractory metals and various forms of carbon have melting points well above $3500K$. Therefore, it is possible for a solar thruster to have an *Isp* of over 1000 *sec*.

There are two basic parameters to a solar concentration system. The first is the total light collection area which is proportional to the input power. The second is the geometric concentration ratio R_c which is the ratio of the collection area to the area of the focused spot. The primary goal of this paper is to show how the propulsion system performs given a particular choice of the above two parameters.

Figure 1 shows a schematic drawing of the solar thruster as considered here. Propellant flows in behind the window and proceeds to the right through the nozzle. Along the way, the propellant is heated in the absorber/heat exchanger which for this paper is a porous disk. The window is required to keep the

* This paper is cleared for public release.

[1] 1. Laug, K., "The Solar Propulsion Concept is Alive and Well at the Astronautics Laboratory", JANNAF Propulsion Mtg., Cleveland Ohio, 1989.

[2] Winston, Roland, "Nonimaging Optics", *Scientific American* 264, March 1991.

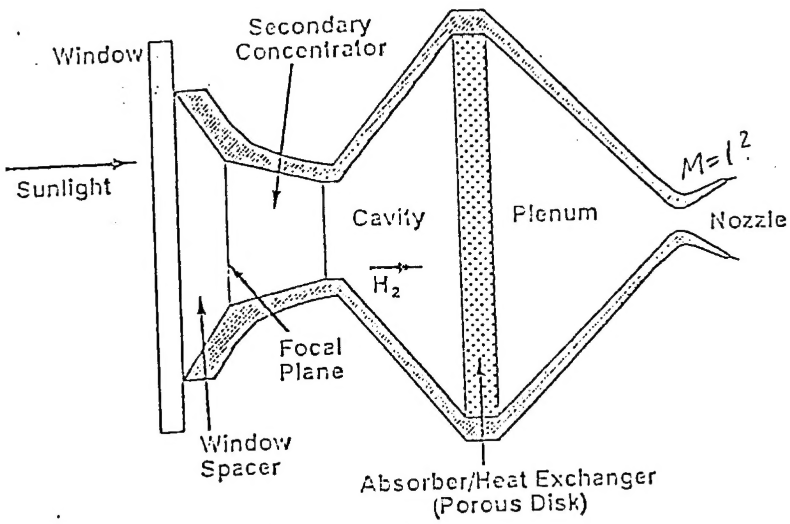


Figure 1: Schematic drawing of solar thruster model.

propellant pressurized. The window spacer spaces the window back from the primary concentrator focal point at the front of the secondary concentrator. This minimizes heating of the window. The window spacer also shields external hardware, bounces some stray rays into the secondary concentrator, and provides some heat to the propellant if the spacer is regeneratively cooled. The secondary concentrator increases the solar flux thereby allowing the aperture area of the absorber/heat exchanger to be reduced. This in turn decreases radiative losses and increases the maximum absorber temperature possible. The secondary concentrator works by maximizing the number of straight paths that connect the cavity to the solar surface. The cavity is a region where sunlight is allowed to expand to a larger area and be absorbed by the absorber/heat exchanger. This expansion allows the surface area of the porous material to be increased thereby increasing heat transfer to the propellant and increasing light absorption. The plenum and nozzle perform the same function as for a conventional rocket.

From the point of view of the Sun, the concentrator of an ideal solar thruster is filled with a view of only the entrance aperture of the secondary concentrator. This aperture will appear to glow at a temperature characteristic of the cavity and absorber/heat exchanger. From the point of view of anywhere within the cavity, the cavity aperture is filled with sunlight. For a real thruster of course there will be losses. The view from within the absorber/heat exchanger might include some cold space due to mirror surface slope errors for example. The window will reflect a small fraction of incident light from either direction. The secondary concentrator will not reflect all incident light. This is only a partial list of the energy losses. However, for this paper, only the ideal thruster is considered to bring out the basic relationships between thrust, Isp, concentration ratio, and collection area.

The Model

In the thruster model presented here, the window is not considered. In a real thruster the main effect of a window is to reflect a fraction of the incident light. At the same time, light from the inside will also reflect back in or be absorbed depending on the wavelength. The window is expected to lower the thrust slightly and possibly increase the Isp. The general behavior of the thruster should not change due to this omission in the model. The window spacer is a cone that converges with the incoming radiation to the aperture of the secondary concentrator. Its main function is to space the window back from the focal point to minimize the window temperature and resulting thermal stresses. It also shields nearby thruster components from stray light.

The secondary concentrator has a concentration ratio defined as the ratio of the exit area to the entrance area. The total system concentration ratio R_c is the ratio for the secondary times the ratio for

the primary mirror. In this model, the concentration at the exit aperture of the secondary concentrator is a given parameter. The particulars of the design of this component are not considered here. It is only assumed that is possible to make the secondary concentrator. In fact, it could be left completely out of any actual thruster at the cost of lower temperature and reradiation losses as will be shown.

The function of the cavity is to make the absorber/heat exchanger to appear as much like a black body as possible. It also increases the surface area of the absorber by increasing its diameter making it both a better absorber and heat exchanger.

The absorber/heat exchanger, which is a porous disk, is where almost all of the numerical work occurs. It is assumed that the thruster will operate in a steady state fashion with all partial derivatives with respect to time equal to zero. Solar energy enters the porous disk from the same side as the propellant. Radiation transfer in the disk is assumed to be a diffusion process with radiative energy traveling in both directions. The propellant travels in only one direction so that energy carried by the propellant travels only towards the nozzle. The diffusion equation for radiation transfer consists of two first order differential equations.

$$\dot{q}_R = -\epsilon\sigma\delta\frac{dT_R^4}{dx}$$

$$\frac{d\dot{q}_R}{dx} + \alpha(T_R - T_H) = 0$$

$\dot{q}_R$ is the net radiative energy flux. It is the intensity of light going to the right minus that going to the left. δ is a parameter related to the mean free path of light through the porous material. In this case it is set to pore size. ϵ is the emissivity of the material that the porous material is made of and σ is the Stefan-Boltzmann constant. The α term in the second equation is an energy sink due to the transfer of radiative energy to the propellant via the porous material. The R subscripts in this paper designate radiation field parameters and the H subscripts designate propellant parameters. The temperature of porous material itself is not directly considered in this paper because we know so little of its properties (this is discussed further below). We are looking for behavior that is independent of the material properties here.

There are four equations describing the propellant behavior.

$$\frac{1}{\rho}\frac{d\rho}{dx} + \frac{1}{v}\frac{dv}{dx} = 0$$ cont

$$\frac{d\dot{q}_R}{dx} + \frac{d}{dx}(c_p\rho v T_H) + \frac{3}{2}\rho v^2\frac{dv}{dx} = 0$$ energy

$$\frac{dp}{dx} = 0$$

$$p - \rho R T_H = 0$$

The first three are statements of conservation of mass, of energy, and of momentum respectively. The fourth equation is the state equation for an ideal gas. These equations represent an initial value problem while the two radiative equations represent a boundary value type problem.

The whole set of six equations are coupled by the $\alpha(T_R - T_H)$ terms. This is the term that transfers radiative energy to heat energy in the propellant. In this term I have lumped together the radiative energy transferred to the porous media and the thermal energy transferred to the propellant. This two step process is considered as a one step process. This is done because the transfer of energy is ultimately between the radiation field and the propellant. The porous media is just a intermediary. Also we do not know the optical or conduction properties of the porous media very precisely so that a more sophisticated transfer model will not help at this time.

The problem represented by the above six equations is broken down into two problems. The boundary value problem is solved first using an initial guess at the propellant temperatures. The initial value problem is then solved given the radiation field temperatures. This two step process is repeated until a solution is converged upon or until it is clear no solution will be found. This is basically a relaxation method. The two radiative equations are solved by solving the N equations that result when the problem is discretized. The other four equations are solved using the Euler method.

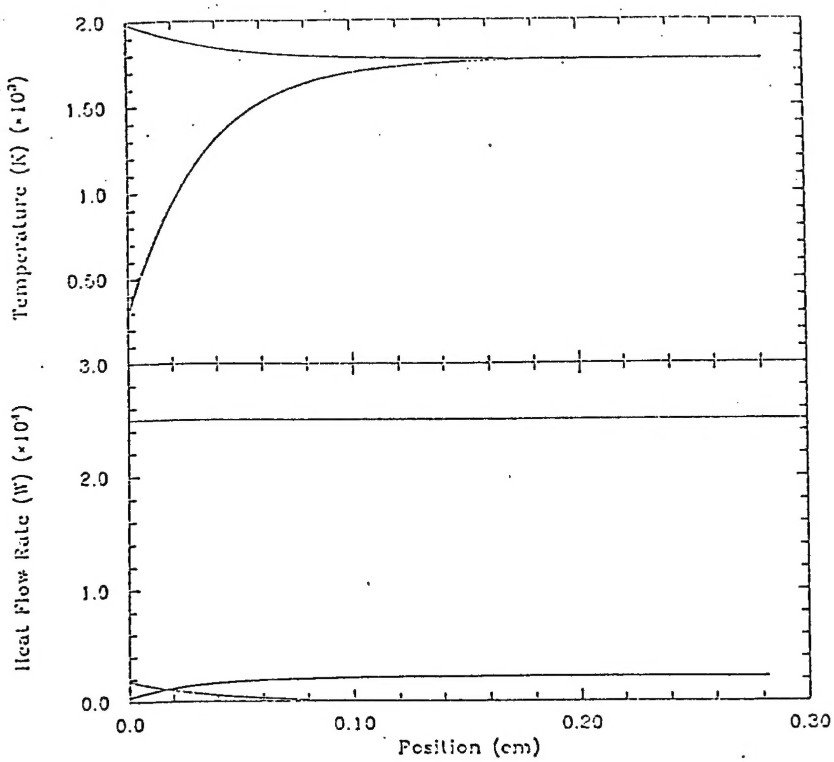


Figure 2: Propellant/Radiation temperatures and heat flow rates as a function of axial position. $R_c = 650$. Power= $25kW$. $\dot{m} = 0.1gm/sec$

Since the combined six equations form a boundary value type problem, boundary values must be specified. The boundary condition at the nozzle side of the absorber/heat exchanger is relatively easy. Conditions at this boundary should not be changing much. If they are, then the chosen porous disk is too thin to function properly. Therefore, the condition for all variables at this boundary is that $\frac{d\phi}{dx} = 0$ where ϕ is any model variable. At the cavity side the boundary conditions for the propellant variables are essentially initial conditions.

The radiation entering the cavity aperture will have an intensity I_c that can be characterized by a temperature T_c such that $I_c = \sigma T_c^4$. If the absorber cavity were actually a gray body at equilibrium temperature T_c, then the energy entering would balance the energy exiting. When propellant is flowing, the absorber/heat exchanger will radiate at a temperature $T_0 < T_c$. In general the diameter of the porous disk is less than the diameter of the entrance to the cavity. The power input will be $I_c A_c$ where A_c is the area of the cavity entrance aperture. In a one dimensional model the power reradiated out of the cavity aperture is $I_0 A_c$ where I_0 is the intensity of light leaving the porous disk surface. The intensity at the porous disk surface is $I_c A_0/A_c + I_0(1 - A_0/A_c)$ and the where A_0 is the area of the porous disk. The emissivity of the porous disk is assumed constant at all wavelengths.

Case 1

Figure 2 shows several model variables as a function of position within the porous disk. The top plot in this figure shows two temperature curves. The bottom curve is the propellant temperature and the top curve shows the temperature characteristic of the radiant flux present at a particular position in the porous disk. The modeled thickness of the porous disk is about 0.3 cm. These two curves meet at a temperature of about $1750K$. At this position in the disk the radiation temperature is in equilibrium with the propellant. The temperature curve for the porous media would lie somewhere between these two curves. At these temperatures the radiative heat transfer rate is about a third of the convective rate. Therefore, the porous disk temperature should lie closer to the propellant curve.

The bottom plot of Figure 2 shows the energy flow for both the radiation and propellant. At $x = 0$

just over 1.8 kW of radiant energy is flowing into the porous disk. At the same position about 300 W of propellant energy is flowing (the propellant starts at about room temperature). The straight line across the top of the bottom plot is at 25 kilowatts, which is the power input.

The convective rate was chosen to be $1kW/m^2/K$. This is on the low side for forced convective transfer and so is a conservative value. Additionally, hydrogen's convective heating rate is an increasing function of temperature. For the purposes of this paper a single value is adequate.

The optical depth of the porous disk is assumed to be three times the mean pore size. The ratio of surface area to volume of a single pore is taken to be one. The emissivity of the porous disk material is set to 0.8 and there are about 40 pores per centimeter. These properties are educated guesses made by examination of porous disks that the Solar Propulsion Group intends to test. The porous disk will remain at a constant temperature so the heat capacity is not needed.

The diameter of the disk is the same as the diameter of the focused sunlight. This diameter is 19 cm and is the diameter of the disks we plan to test. The total power focused is 25 kW which is what our terrestrial 7.7 × 6 meter concentrator should focus. This concentrator would have to be defocussed somewhat to get a spot size this large. The resulting geometric concentration ratio is about 650. In this particular case the secondary concentrator is nonexistent and the cavity is a straight tube. This is about as simple as this model can be.

Modeling of the thruster nozzle is done by looking up the Isp from a table calculated using hydrogen properties [3] . For the case illustrated by Figure 2, the Isp is about 5900 m/sec(600 sec). The mass flow rate in this case is 0.1 gm/sec so the thrust is about 0.59 N(0.13 lbf). The actual amount of energy added to the propellant is about 7.3 percent of the incident solar radiation or about 1800 watts. Therefore, operating in this mode is not particularly efficient nor does it provide much in the way of Isp.

From the information in the above paragraphs it should be apparent why we are doing this modeling. We have recently modified a calorimeter to look much like this case. Comparison of the laboratory version of this case will allow us to improve on our model. In particular, we should be able to narrow down some of the values that we are guessing at, such as convective heat transfer rate.

Case 2

Figure 3 shows a case identical to the previous case except that the 25 kW are put through a 8 cm diameter spot instead of a 19 cm spot. This results in a concentration ratio of 3700 (which is close to the concentration ratio that our facility is producing currently). The cavity expands to a 19 cm diameter to match our porous disks. The final propellant temperature as viewed from Figure 3 at $x = 0.28$ cm is about 2900K. This corresponds to an Isp of about 8800 m/sec(900 sec) and a thrust of 0.88 N(0.2 lbf). The efficiency of this version of a thruster is about 12.5 percent. Note that the radiative flux (decreasing curve in bottom plot of Figure 3) is zero by 0.2 cm into the porous disk. This means that about 87.5 percent of the incident flux finds its way back out of the cavity aperture.

The concentration was increased by about 5.7 in this case over the last case. This ratio is proportional to the intensity increase at the cavity aperture. Additionally, the efficiency of heat transfer to the propellant is 1.71 times higher in this case. The ratio of the propellant temperature changes in the two cases is about 1.73. The fourth power of 1.73 is about 9 which is about equal to 5.7 times 1.71. Therefore, the temperature of the propellant increases approximately as the characteristic radiation temperature times the fourth root of any efficiency increases. The Isp in turn goes approximately as the square root of the propellant temperature. The Isp increases at better than the eighth root of the geometric concentration ratio increase.

Case 3

Figure 4 shows a case with the same parameters as the previous one except a secondary concentrator is added which increases the concentration ratio by a factor of three to about 11000. The final propellant temperature in this case is about 3800K. This temperature corresponds to an Isp of about 11,800 m/sec(1200 sec) and a thrust of 1.2 N(0.26 lbf). Seventeen percent of the incident radiation is absorbed into the propellant. The increases in temperature and Isp are consistent with the ratios

[3] King, C. R., "Compilation of Thermodynamic Properties, Transport Properties, and Theoretical Rocket Performance of Gaseous Hydrogen", NASA Technical Note D-275, April 1960.

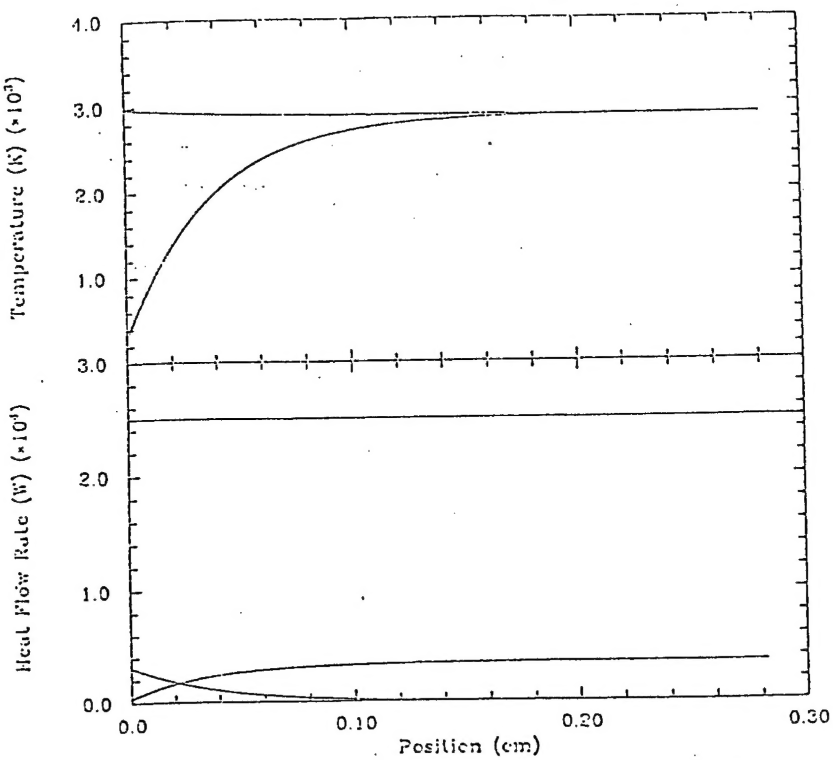


Figure 3: Propellant/Radiation temperatures and heat flow rates as a function of axial position. $R_c =$ 3700. Power= $25kW$. $\dot{m} = 0.1gm/sec$

for the previous cases. The Isp increases at better than the eighth root of concentration ratio and the temperature increases at about the fourth root of the concentration ratio.

Case 4

Figure 5 shows the results from the fourth case. This case simply doubles the mass flow rate of the previous case. The result is a lower temperature, $T = 3300K$, a lower specific impulse $Isp =$ 9800 m/sec(1000 sec), a nearly doubled thrust $F = 2.0$ N(0.44 lbf), and a nearly doubled efficiency of 30 percent. Another case was run (but not plotted here) with a mass flow rate five times the previous case. This rate increased the efficiency to 43 percent, the Isp decreased to 6900 m/sec(700 sec), and the thrust tripled to 3.43 N(0.8 lbf).

Also notice in Figure 5 that the radiative flux (decreasing curve of bottom plot) stays nonzero longer than in the previous case. This is because the temperature is lower towards the back of the porous disk than in the previous case. Radiation is encouraged to diffuse further to the back. Perhaps this is what the term "radiation trapping" refers to. This deeper diffusion does result in higher efficiency but at a cost to propellant temperature.

Case 5

The final case is shown in Figure 6. This case is meant to represent what might happen to a thruster at a distance from the Sun twice that of the Earth. To simulate this, the radiated power is reduced to a fourth, the aperture diameter is reduced to half, and the propellant flow rate is reduced to a fourth. This simulates the $\frac{1}{r^2}$ radiation decrease and the angular size of the sun halving. Presumably only a quarter of the propellant could then be heated to maintain the same thrust. All other parameters are the same as in the third case.

Figure 6 shows the propellant temperature reaches about $3800K$ which is essentially the same temperature as in Figure 4. The Isp will be about the same and therefore the thrust will be one quarter that in the third case since the propellant flow rate is one quarter. Therefore, Isp can be maintained at

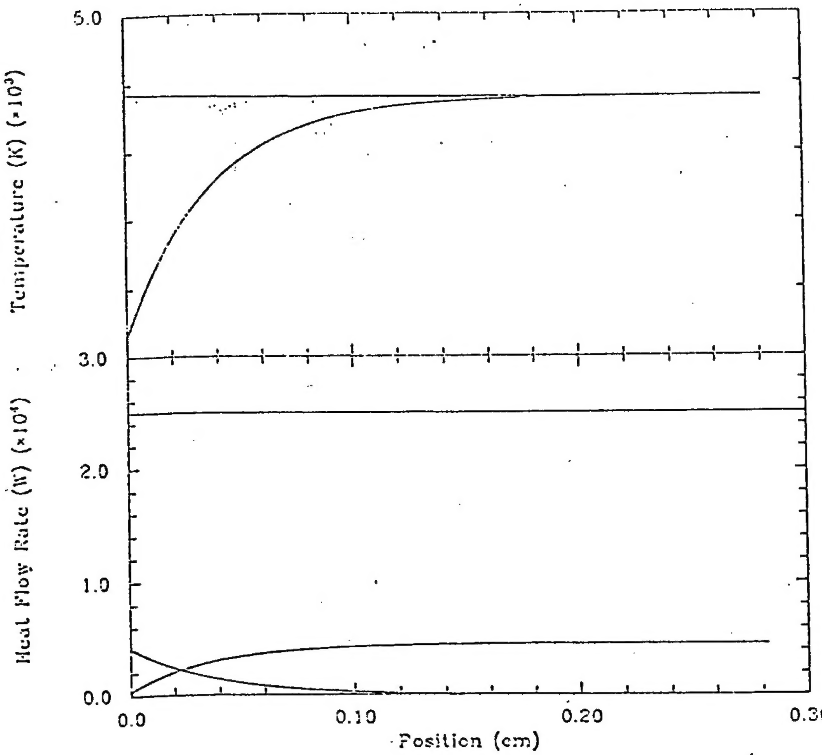


Figure 4: Propellant/Radiation temperatures and heat flow rates as a function of axial position. $R_c =$ 11000. Power= $25kW$. $\dot{m} = 0.1gm/sec$

further distances from the sun. The price is paid with decreased thrust.

Discussion

There are two rules of thumb that are confirmed or improved upon in this study. The thrust is approximately proportional to input power and the specific impulse Isp increases faster than the concentration ratio to the eight root. This last rule is somewhat suspect since the internal energy modes of the propellant were not considered in the heat exchanger as they were in the Isp calculation [3].

Solar rockets can work outside of Earth's orbit. They can even go to Pluto if you are willing to work with millinewtons of thrust at Pluto (and thermal losses don't take too much energy).

The cases discussed here showed rather low efficiencies. However, the efficiency could be increased by adjusting parameters. A larger cavity diameter would probably increase the efficiency since this decreases the propellant flow rate. This in turn increases the residency time at the hot areas and increases the heat flow into the propellant. The choice for cavity here was chosen to match actual Phillips Laboratory Solar Laboratory hardware.

Much is lacking in this model, partly because of a lack of data and partly because of a lack in time for improvements. Nonetheless, it does the right things. It shows the effect of changing concentration ratio and of changing geometry. It is also insensitive to the optical properties of the heat absorber/exchanger. This can either be good or bad depending on your point of view. In our case we do not know the needed optical properties so this is good. We aren't fooled into thinking we know more than we really do. When we perform our porous disk experiments we will learn much from comparisons to this model. For example, if the disk temperature tracks the radiation curve more closely than the propellant curve, then the actual convective heat transfer coefficient is lower than that used here. If this disk temperature follows the propellant temperature closely, then the convective heat transfer coefficient is higher than expected.

[3]King, C. R., "Compilation of Thermodynamic Properties, Transport Properties, and Theoretical Rocket Performance of Gaseous Hydrogen", NASA Technical Note D-275, April 1960.

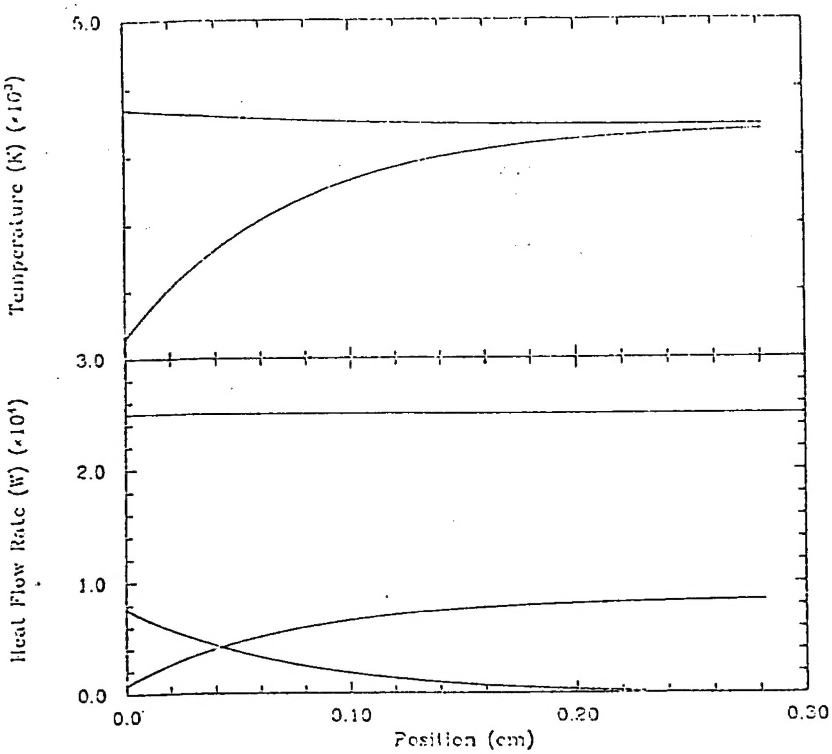


Figure 5: Propellant/Radiation temperatures and heat flow rates as a function of axial position. $R_c =$ 11000. Power$= 25kW$. $\dot{m} = 0.2gm/sec$

Thermal and reflectance losses will of course affect the ultimate performance. A secondary concentrator will probably transmit about 90 percent of incident light. It can be regeneratively cooled to make up for this loss. The conductive losses through the thruster walls are unknown at this point, but could be minimized given how thin the porous disk is. The thruster can be very short thereby reducing the surface area.

This particular thruster design uses the best features of several popular solar thruster concepts. It uses a cavity to increase the absorber emissivity. It is a volumetric type absorber so that heat does not need to flow through a wall before being absorbed. The absorber/heat exchanger only sees the pressure drop required to produce the proper mass flow. As a result, the porous disk can be loosely placed which will allow for expansion and contraction. The porous disk can be very thin, thus reducing the thermal stresses. On the other hand, the disk could be made considerably thicker to allow for loss of pore filaments due to thermal stress induced breakage. This should have a minimal effect on the propellant temperature.

High solar concentration appears to be a necessity to get efficiency, thrust, and Isp to high levels. It is probably necessary to have a secondary concentrator to supplement the primary concentrator. Error free primary concentrators can obtain a concentration ratio of 11,000 at most. The cases that performed best had a total concentration ratio of about 11,000. It is unlikely that an inflatable concentrator will achieve this concentration on its own.

Future Work

Unfortunately, there may not be further work in solar propulsion at the Phillips Laboratory. Manpower and budget cuts may result in this project being cut. Presumably, lower cost access to space is a high priority in this environment of shrinking resources. Solar propulsion and other advanced concepts could conceivably save their development costs within a few launches. Additionally, solar propulsion could be enabling for interplanetary missions.

Assuming the best, we intend to test the porous disk absorber/heat exchanger concept in the labo-

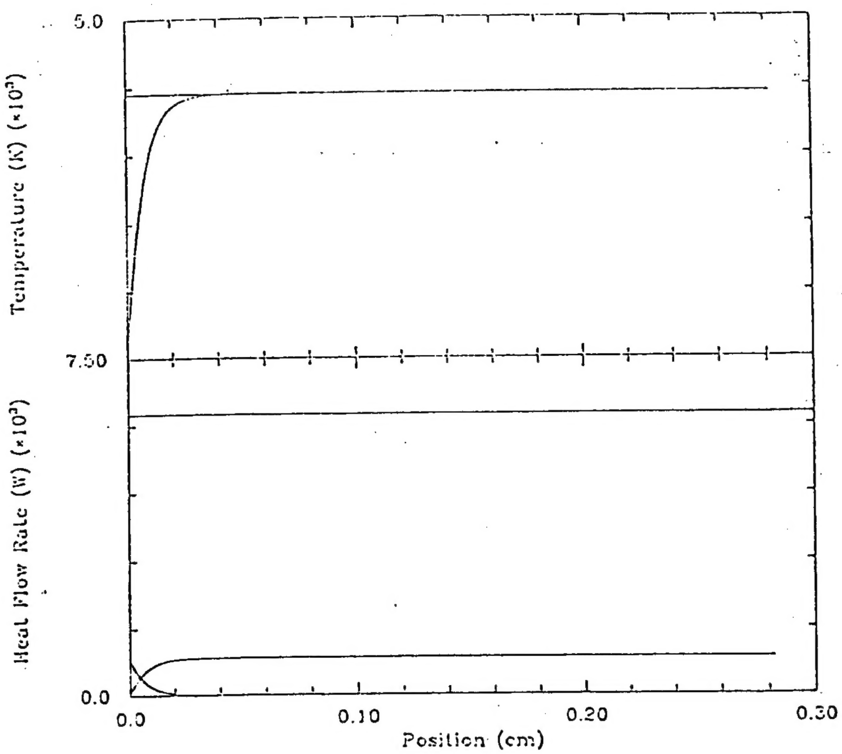


Figure 6: Propellant/Radiation temperatures and heat flow rates as a function of axial position. $R_e = 11000$. Power= $6.25kW$. $\dot{m} = 0.025gm/sec$

ratory. If these experiments work well, improvements will be made to this model. In addition, there are modeling efforts proceeding on future and past thruster concepts by us and others. We also hope to test outside thruster designs in our facilities.

One Dimensional Model of a Solar-Thermal Thruster Using a Porous Absorber/Heat Exchanger

Michael R. Holmes
PL(OLAC)/RKCO

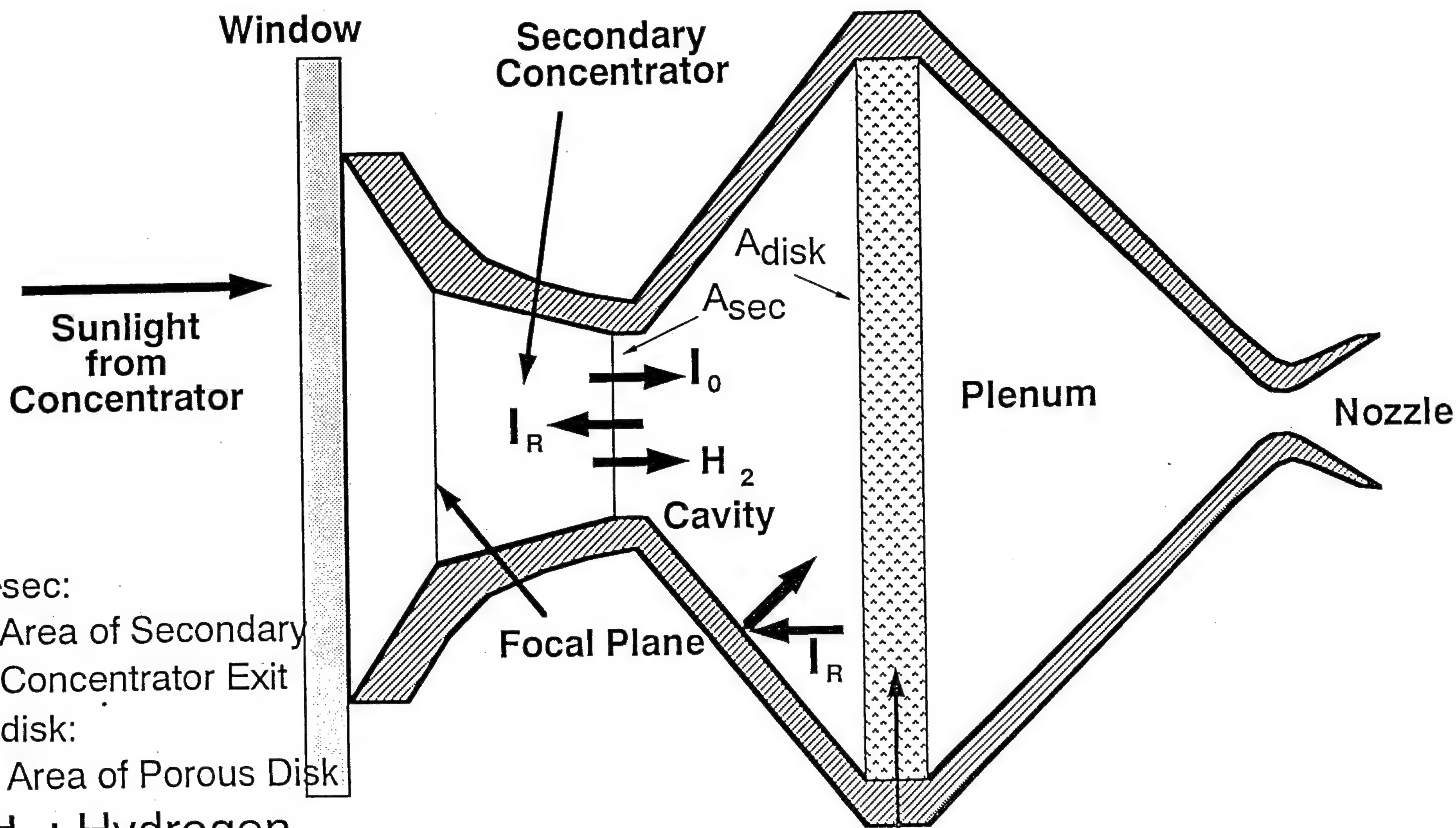
Purpose

- Implement 1st Order Physics
- Demonstrate 1st Order Thruster Performance
- Model to be Compared With Experiment (RVCT)
- Find Performance (Specific Impulse, Isp; Thrust, F) as a Function of Geometric Concentration ratio Rc

The Model

- Window Perfectly Transparent
- Concentrator Flux Given at Secondary Exit
- One Dimensional Energy and Hydrogen Transport
- Radiant Energy Obeys Diffusion Equation
- Heat Transfer to Hydrogen Proportional to Temperature Difference Between Hydrogen and Radiation Field
- Output is Hydrogen Temperature and Approximate Isp

The Model (Schematic)



A_{sec}:
Area of Secondary Concentrator Exit

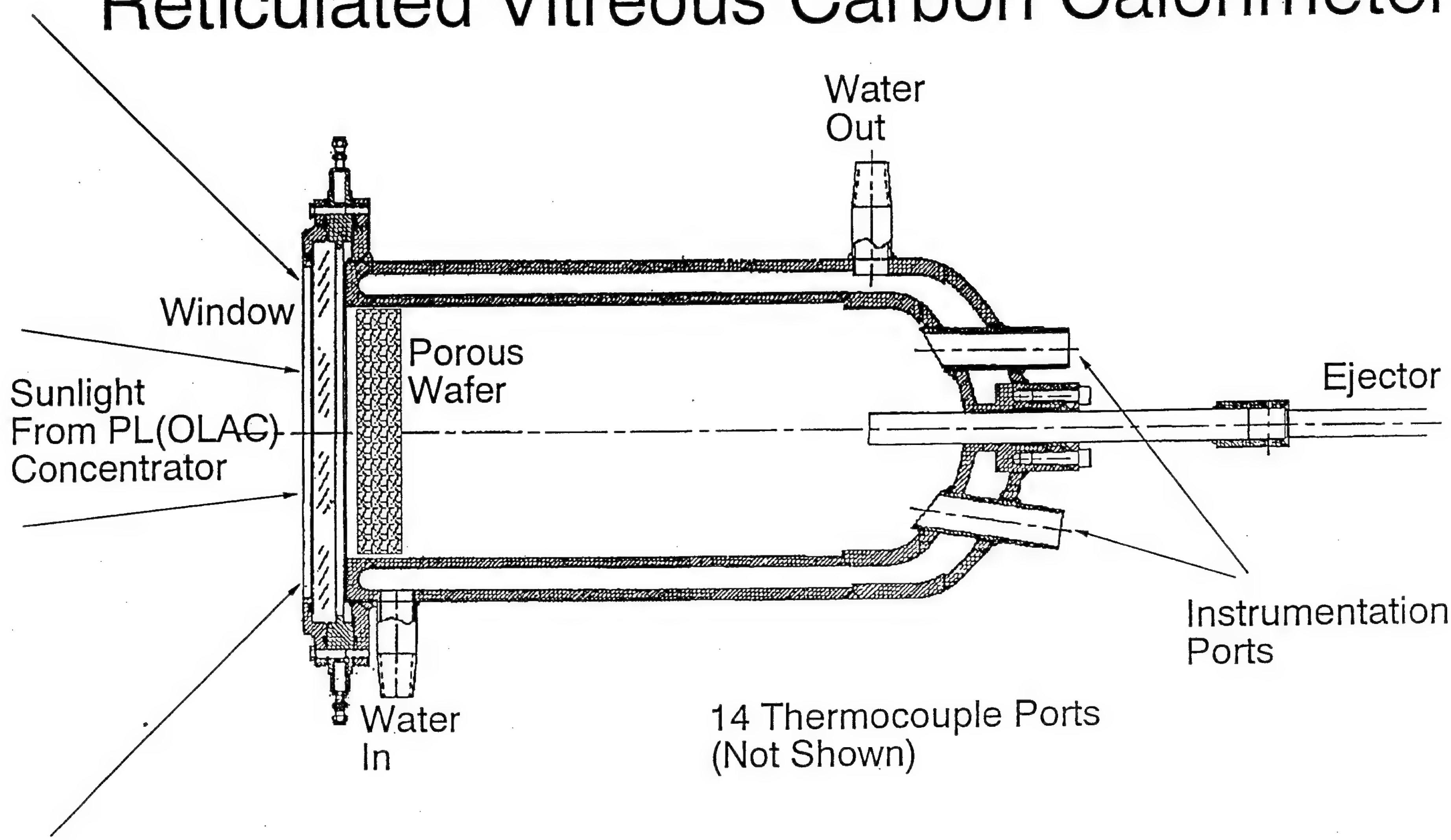
A_{disk}:
Area of Porous Disk

H_2: Hydrogen

I_0: Incident Flux

I_R: Reflected and Reradiated Energy

RVCT
Reticulated Vitreous Carbon Calorimeter



Equations

$$\dot{q}_R = -\epsilon\sigma\delta\frac{dT_R^4}{dx}$$

$$\frac{d\dot{q}_R}{dx} + \alpha(T_R - T_H) = 0$$

Diffusion

Radiation Transport (Boundary Value Problem)

Coupled by α

$$\frac{1}{\rho}\frac{d\rho}{dx} + \frac{1}{v}\frac{dv}{dx} = 0$$

Continuity

$$\frac{d\dot{q}_R}{dx} + \frac{d}{dx}(c_p\rho v T_H) + \frac{3}{2}\rho v^2\frac{dv}{dx} = 0$$

Energy

$$\frac{dp}{dx} = 0$$

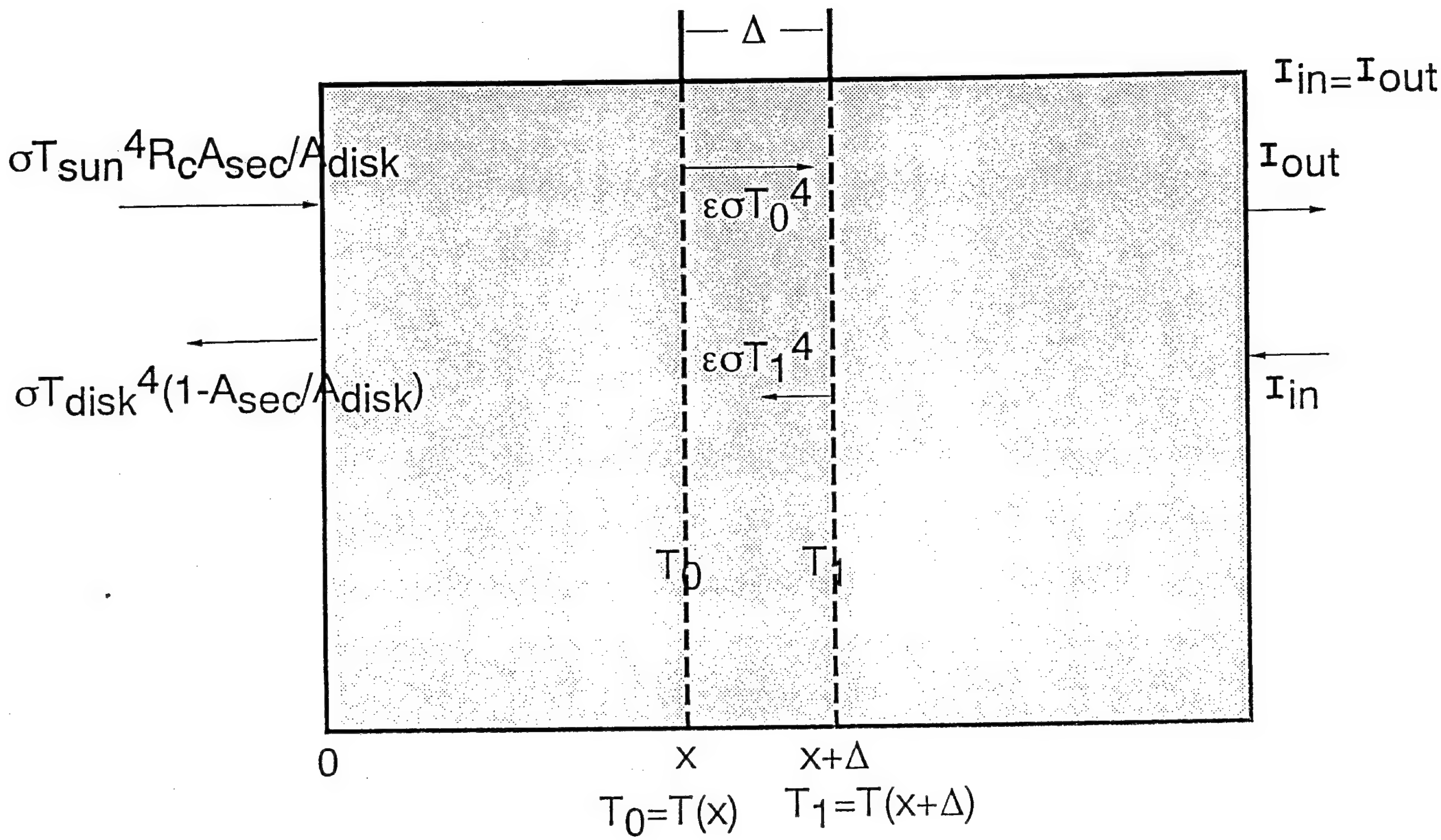
Momentum

$$p - \rho R T_H = 0$$

Eqn. of State

Hydrogen Flow (Initial Value Problem)

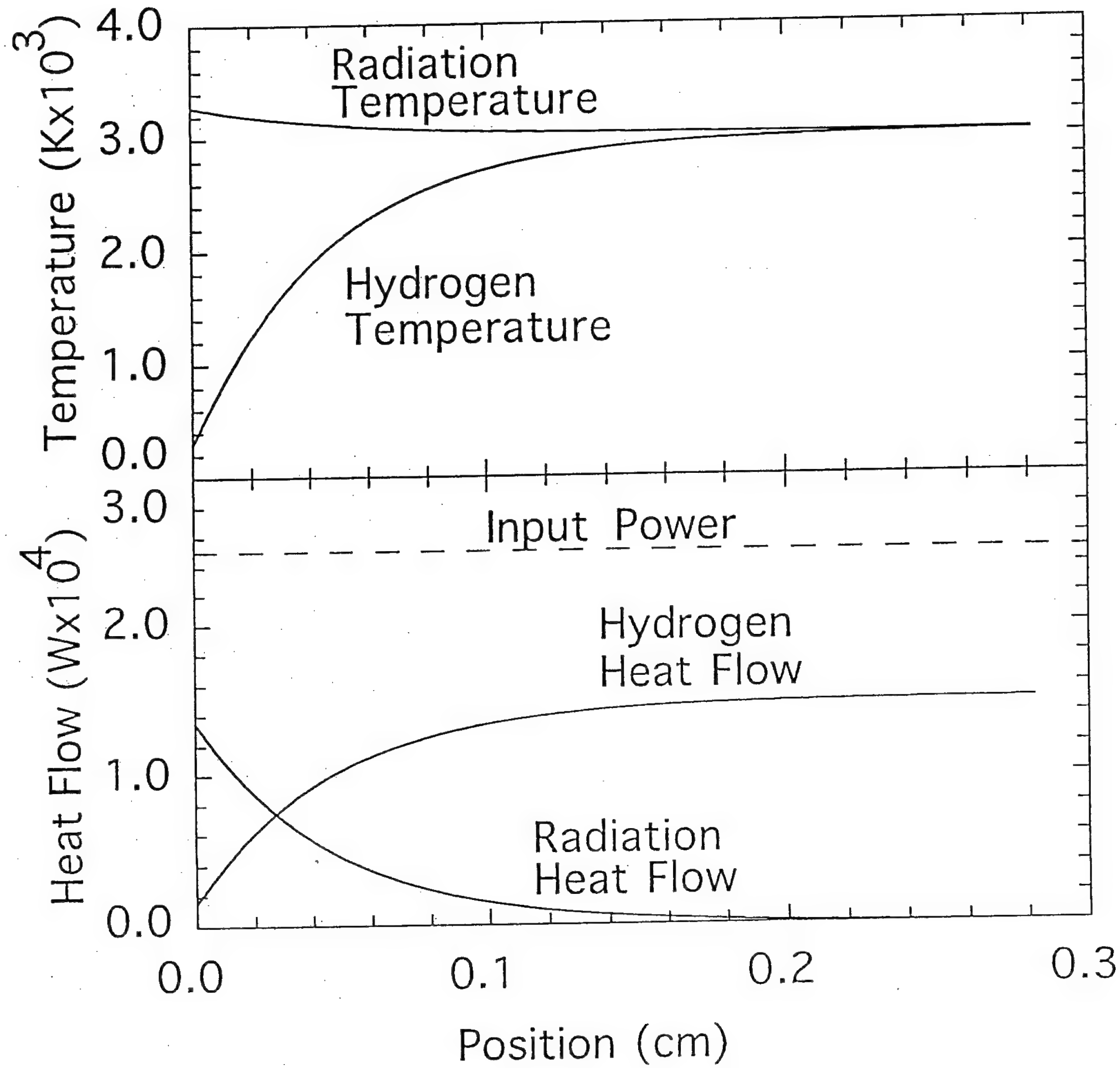
Radiation Transport in Disk



Numerical Solution

- Equations are Discretized
- Initial Guess of Variable Values on Grid
- Repeat Until Self Consistent Temperatures:
  - Solve Boundary Value Equations (Radiation)
  - Solve Initial Value Problem (Hydrogen)
- Plot Results

Example Case



5 Sample Cases

Case 1: Simulates RVCC Experiment; 19 cm Aperature

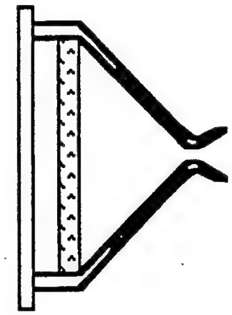
Case 2: Same as Case 1 Except 8 cm Aperature

Case 3: Same as Case 2 With Secondary Giving 3X Concentration

Case 4: Same as Case 3 With Doubled Mass Flow

Case 5: Operation at Twice Earth's Distance From Sun

Case 1

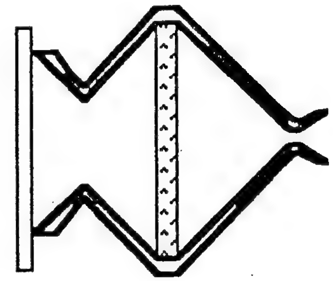


0.1 g/sec: P=25kW:
Rc=600

F=0.6N or 0.13 lbf
T=1750K or 2600°F

Isp≈600sec

Case 2

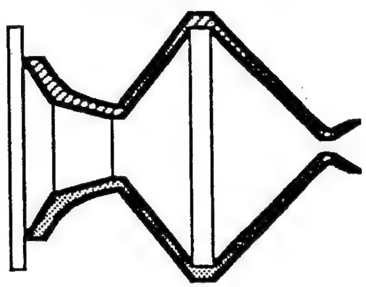


Rc=3700

F=0.9N or 0.2 lbf
T=2900K or 4700°F

Isp≈900sec

Cases 3 & 4



Rc=11000
0.1 g/sec & 0.2 g/sec

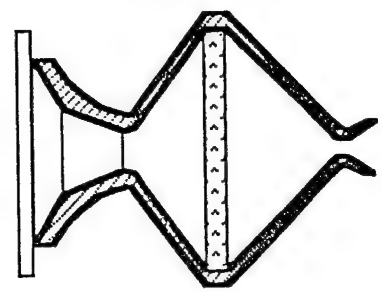
F=1.2N or 0.26 lbf
T=3800K or 6300°F

Isp≈1200sec

F=2.0N or 0.44 lbf
T=3300K or 5400°F

Isp≈1000sec

Case 5



P=P/4: Rc=11000:
0.025 g/sec

F=0.3N or 0.065 lbf
T=3800K or 6300°F

Isp≈1200sec

Conclusions

- $Isp \propto Rc^{1/8}$
- $F \propto P$, Power from Concentrator
- Everything Interesting in Thin Layer (a Few Mean Free Paths)
- Isp Maintainable with Distance From Sun
- Secondary Concentrators and Cavities are Good
- Large Surface Area Within Thin Layer Gives High Efficiency

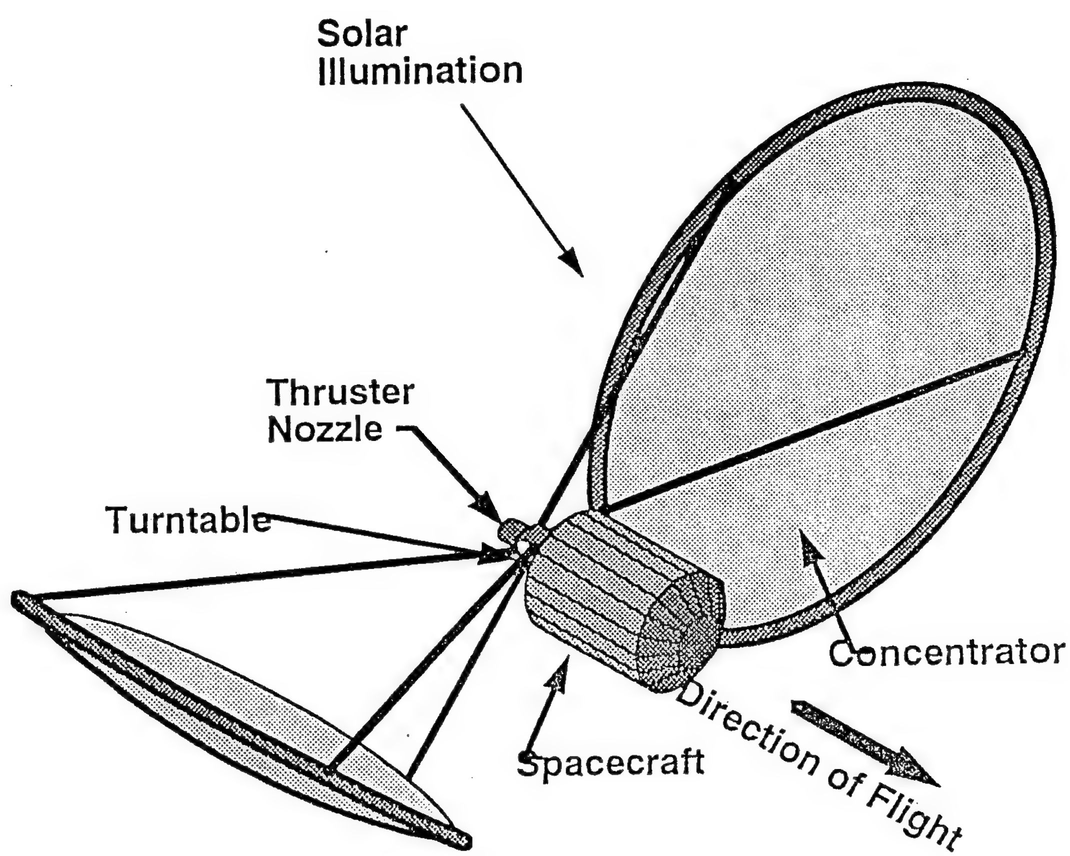
Solar-Thermal Thruster Design Notes

- Secondary Concentrators are Probably Necessary for High Isp
- If Reflective Surfaces are not Possible on Secondary, Small Entrance Aperature is Desirable
- Thick Disks will Allow Thruster to Function Well Even after Deterioration of Disk at Front Face
- Cavity Absorbers Allow Larger Disk Surface Area and Less Reradiation
- Window will Lower Input Power But Not Necessarily Temperature.
- Porous Media Is Not a Supporting Structure

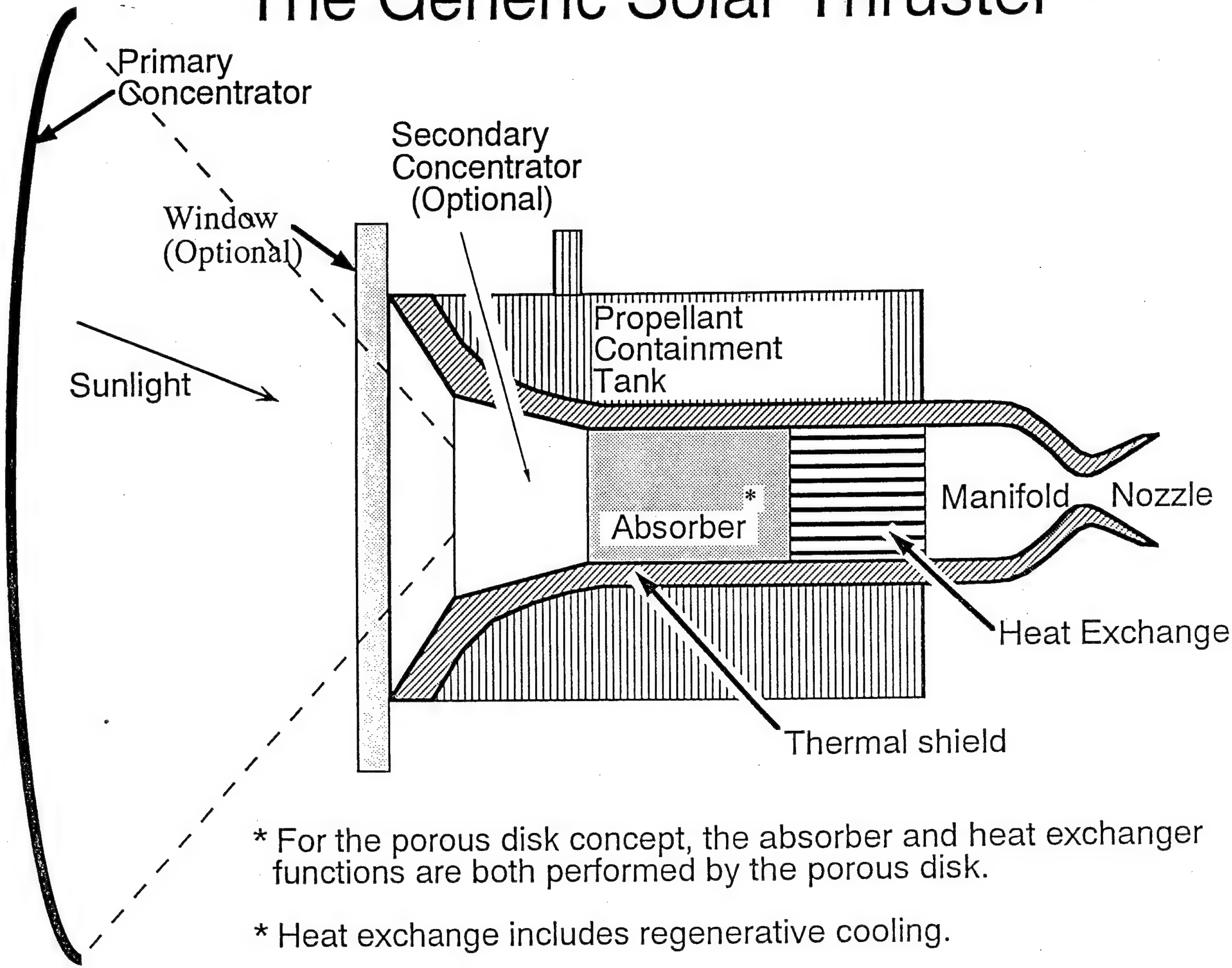
Appendix

What is Solar-Thermal Propulsion

Solar-Thermal Spacecraft Concept



The Generic Solar Thruster



* For the porous disk concept, the absorber and heat exchanger functions are both performed by the porous disk.

* Heat exchange includes regenerative cooling.

Why Solar-Thermal Propulsion?

- Specific impulse is about double current upper stage thrusters, i.e., double the payload.

- Conventional rockets use chemical reactions to generate energy. The reaction byproducts then become the propellant. The molecular mass of these byproducts limits Isp.

- Solar rockets use sunlight available in orbit as an energy source. Therefore, a more efficient propellant, hydrogen, is used.

Who is Working Solar-Thermal?

OLAC Phillips Lab

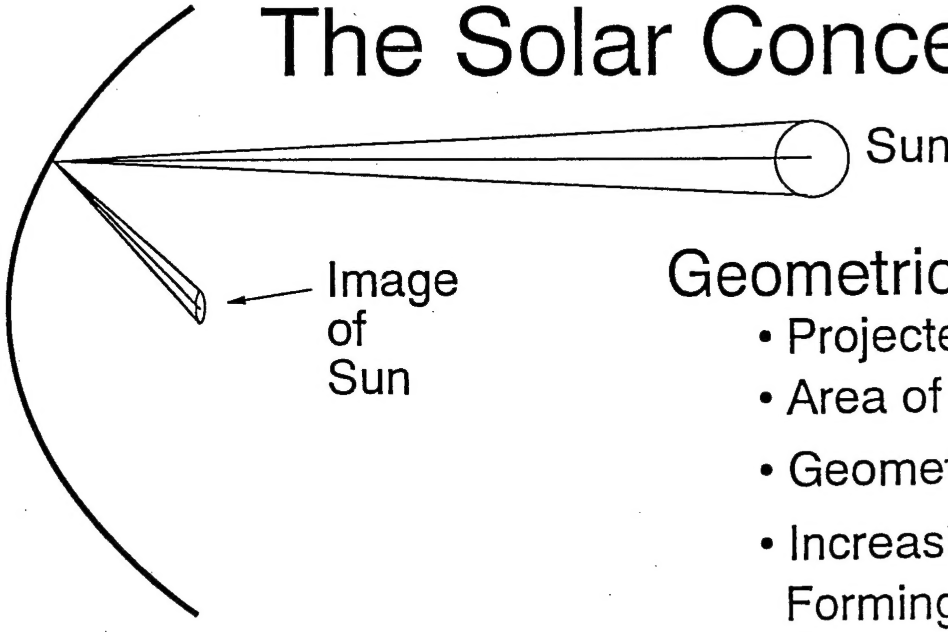
Consortium

- UAH
- PL/RKCO
- NASA - MSFC
- NASA - LeRC
- SRS Technologies
- Rocketdyne
- McDonnell-Douglas Aerospace

NASA - MSFC

Hercules

The Solar Concentrator



Geometric Concentration Ratio

- Projected Area of Concentrator=A_C
- Area of Image=A_{sun}
- Geometric Concentration Ratio $R_C \equiv A_C/A_{sun}$
- Increasing Rc Increases Solid Angle of Light Forming Image

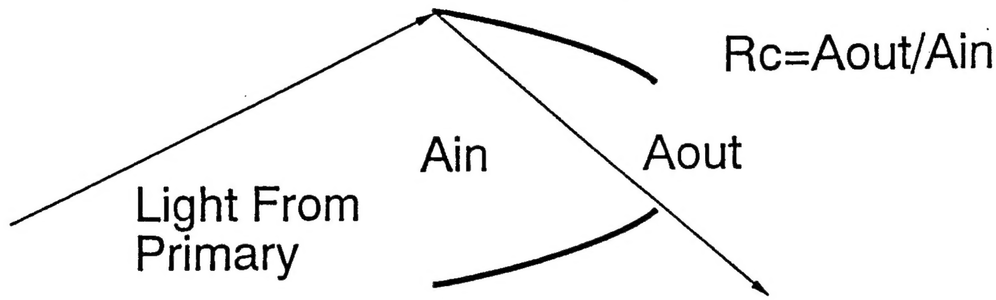
Thermodynamic Limit

- Max Temperature is Temperature at Surface of Sun
- Max Temperature Corresponds to $R_C \approx 46{,}000$
- For $R_C = 46{,}000$; Solid Angle=2π Steradians

Parabolic Concentrator

- Preferred Concentrator Shape
- Focuses All Rays Parallel to Axis of Symmetry to a Single Point, Defined as the Focal Point
- High Concentration Ratios Give Poor Images
- Practical Limit for $R_C \approx 11{,}500$

The Secondary Non-Imaging Concentrator



- Further Concentrates Light From Primary by Factor Aout/Ain
- Can give Rc approaching Thermodynamic Limit of 46,000 in Conjunction With Parabolic Concentrator of Long Focal Length
- Limits Reradiation by Reducing Aperature of Absorber (Physically Equivalent Statement as Previous Bullet)
- Destroys All Image Information
- Angular Spread of Exiting Light Increases
- Must Operate at High Temperature

Solar-Thermal Thruster Performance Upper Limits

In the limit where the propellant carries away negligible energy;

$$T^4 = \frac{R_c I_{sun}}{\sigma} \quad \text{Stefan-Boltzmann}$$

Specific impulse Isp follows;

$$Isp \propto \sqrt{\frac{T}{m}}$$

Therefore, max possible Isp follows;

$$Isp \propto R_c^{1/8}$$

Thrust is given by Newton's Law, F=dm/dt v, where v is the propellant velocity, and dm/dt is the propellant mass flow rate. Power put into propellant flow is Fv or dm/dt v^2. v^2 is proportional to the kinetic energy imparted to the propellant.

If the Isp is held fixed, then v is also constant and thrust F is proportional to the area of the concentrator.

If F is held constant, then v is proportional to the area of the concentrator and dm/dt is inversely proportional to v. Isp is proportional to v and also increases. Therefore, thruster temperatures must also increase.

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